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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

Flight envelopes and off design engine performance for several subsonic nuclear cruise aircraft are studied. Each airplane weighs one million pounds and uses a helium-cooled thermal reactor with turbofan engines. Cruise design points of Mach 0.3 and 2000 feet, Mach 0.55 and 20 000 feet, and Mach 0.8 and 40 000 feet are considered. The Mach 0.55, 20 000 foot airplane seems attractive in that it offers a high payload and a fairly large flight envelope. The other two aircraft suffer severe payload or flight envelope restrictions and may be less adaptable for a multipurpose role.

COMPARATIVE FLIGHT ENVELOPES FOR THREE DIFFERENT DESIGN POINT SUBSONIC NUCLEAR AIRPLANES (U)

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SUMMARY

Flight envelopes and off-design engine operating conditions for three different subsonic nuclear cruise airplanes were studied. Each airplane had a ramp gross weight of one million pounds. For propulsion, the airplane used a long-lived (10^4 hrs) thermal reactor with helium/turbocirculator cooling in conjunction with chemically augmented turbofan engines. The three aircraft studied, which differed primarily in powerplant size and in wing configuration, were based upon design-point cruise conditions of Mach 0.3 and 2000 feet, Mach 0.55 and 20 000 feet and Mach 0.8 and 40 000 feet. The intermediate design-point airplane appears to be attractive in that it offers a large payload (155 500 lb) in combination with a fairly large flight envelope. In comparison with this, the slower airplane yields a slight payload improvement but is severely limited in its Mach number - altitude capabilities; the faster airplane obtains a slightly better flight envelope at the cost of a 72 percent payload reduction.

Engine operation is found to not differ significantly from contemporary large-turbofan practice. The chemical augmentation, which was necessary because of takeoff requirements, proves very effective also in temporarily enlarging the flight envelopes.

INTRODUCTION

For specialized applications, the nuclear airplane is attractive because it offers almost unlimited range and endurance. Of particular interest is its good cargo-carrying capability at very long range.

Recent studies such as references 1 and 2 have presented the payload capabilities of large subsonic nuclear cruise airplanes having helium thermal reactors and turbofan engines. These studies determined nuclear powerplant sizes on the basis of design point cruise conditions. They assumed that takeoff and climb were on chemical fuel (JP4) only. Adequate fuel was also provided for an emergency chemical cruise and letdown range of 500 miles. Chemical and/or nuclear mode of operation

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was accomplished by placing a heat exchanger in tandem with a conventional burner between the compressor exit and inner turbine entrance. This greatly increased achievable turbine inlet temperature. In reference 1, equations were developed for aircraft structure weight, propulsion systems weights, and nuclear component weights. A combination of twelve independent variables describing the system were simultaneously optimized by a computer program to maximize design point payload for a fixed gross weight.

Reference 1 showed that nuclear powered airplanes become attractive at large gross weights ($\geq$ 1 000 000 lb); that payload fraction increased with gross weight; and that shield technology advances could increase payload by as much as 50 percent.

The effect on payload of lifetime requirements for the reactor and heat exchangers was shown in reference 2. In this reference, it was also shown that lifetimes on the order of 10 000 hours require the use of a new fuel pin concept which is treated in detail in reference 3.

Philosophies with regard to the design and operation of subsonic nuclear aircraft were presented in reference 4. The main emphasis was safety, technical feasibility and practicality in that routine maintenance, handling, and normal operation should not be more difficult than that of conventional large aircraft. The chemical mode of operation was included in the design to enhance safety. As will be shown, this chemical capability greatly increases the versatility of the aircraft.

Although a significant amount of nuclear aircraft performance data reflecting current technology is available for design point conditions, there is by contrast no data for off-design operation. This is a serious lack because a project as costly and complicated as a nuclear aircraft may not be economically justifiable on the basis of one single class of missions. In addition, takeoff and acceleration performance must be examined to ensure that the airplane is capable of reaching cruise (design point) conditions.

Thus it is clearly necessary to study the full range of any given airplane's performance capability and also to define the proper design conditions for a multipurpose aircraft.

As an initial step in this direction, flight envelopes (i.e., range of Mach number and altitude) for nuclear airplanes are studied in the present report. This is done by designing, with the method of reference 1, three one million pound aircraft. These are: a low Mach number (0.3) low altitude (2000 feet) aircraft; one for intermediate Mach number (0.55) and intermediate altitude (20 000 feet); and the last for high Mach number (0.8) and high altitude (40 000 feet) design conditions. These aircraft are shown in figures 1, 2, and 3 respectively.

Since takeoff and climb are on chemical fuel and the nuclear powerplant is sized for cruise operation, it is unlikely that a large range of off-design point capability on nuclear power alone is possible. Therefore, for determining flight envelopes, chemical augmentation is used to raise the turbine inlet temperature to the required level above that obtainable from nuclear power. The envelopes are then functions of the chemical fuel consumption rate at the off design points. Since "payload" (which may consist of some additional chemical fuel) is known for each airplane, the time of operation at each point in the envelope can be determined.

ANALYSIS

Assumptions

The propulsion system assumed for the nuclear airplane is shown schematically in figure 4. With the burner in tandem with the heat exchanger as shown, turbine inlet temperature can be raised above that obtainable on nuclear power alone. For the purpose of this study, a lifetime of reactor, heat exchangers, and engines of 10 000 hours are chosen. As shown in reference 2, this requires the use of the pin and tube reactor concept. The shield material chosen was a heavy metal-water layered shield.

Each engine has associated with it a turbocirculator to pump the helium working fluid. (In a turbocirculator, the helium is expanded through a turbine which is used to drive the helium compressor.) This choice of a turbocirculator rather than an engine driven pump was made to: eliminate the seal problems of an engine driven pump and lower the pressure and temperature of the helium entering the heat exchangers. The combination of lower pressure with lower temperature does result in lower weight for the heat exchangers at long liftimes. Since the temperature going into the heat exchanger is still above "optimum" turbine inlet temperature, this does not hurt performance.

Engines were sized to give 300 feet/minute climb capability at the design point (cruise) conditions. This was to allow for maneuvers and operation on a non-standard day. Fixed nozzles were assumed, and conventional turbines, fan, compressor and combustor were used. Design adiabatic efficiencies for the fan, compressor, and inner and outer turbine were 0.88, 0.86, 0.90, and 0.90, respectively. The nozzle thrust coefficients were 0.975 and the inlet pressure recovery was 1.

Procedure

The airplanes were designed and the thrust required at off-design conditions was calculated as per reference 1. The instantaneous weight of the aircraft was taken to be the initial weight of 1 000 000 pounds minus the takeoff and climb chemical fuel requirements.

To obtain engine performance at off-design conditions, it was necessary to match engine components. Matching is accomplished by conserving mass and energy at corresponding rotational speeds. The procedure is described in detail in reference 5.

The computer code of references 6 and 7 was modified to perform matching of components when a heat exchanger is placed in tandem with the chemical burner. This code requires that maps of the components (i.e., performance) be loaded as data and requires a design point on the maps. Therefore, to accomplish the matching, two things were necessary. The first was calculating off-design performance of the heat exchanger. The second was the use of generalized maps so that the design point on the maps corresponded to the design conditions of the engine previously calculated.

Heat exchanger dimensions, flow areas, etc. were calculated at the design point. Knowing the power (heat load), maximum allowable wall temperature, and helium flow rate, Henry Putre of the Lewis Research Center wrote a computer subroutine to perform the necessary off-design calculations.

The generalization of component maps and incorporation of the heat exchanger subroutine into the matching code was then accomplished with the collaboration of Robert W. Koenig, also the Lewis Research Center. The generalized maps used for the fan and compressor are shown in figures 5(a) and (b). (Similarly, generalized maps for the turbines were used, but are not shown.) Constant percent of design corrected speed lines ($N/\sqrt{\theta}$) are plotted versus percent of design pressure rise ($\Delta P/P$) and percent of design corrected total airflow ($W\sqrt{\theta}/\delta$). In addition contours of percent design efficiency (η) and the surge line are also indicated. Thus given any design point, a specific map for that particular engine can be generated.

With the generalized maps, thrust and specific fuel consumption can be obtained at any off-design point as a function of turbine inlet temperature. By cross-plotting the thrust and finding where it equaled the drag, chemical fuel requirements were determined and flight envelopes developed.

RESULTS AND DISCUSSION

Aircraft Configuration

The important parameters describing the design airplane and its engines previously shown in figures 1 - 3 are tabulated in table I. It is apparent that the low Mach number and altitude (M-H) (fig. 1) and intermediate M-H aircraft (figure 2) are very similar. The major difference between them is the larger wing span of the low M-H aircraft. This larger span results in heavier structure weight and is caused by the greater lift surfaces required at the lower Mach number. In table I the powerplant weight for the intermediate M-H airplane is heavier than that for the low M-H airplane. Although the lift to drag ratio at cruise is higher for the low M-H aircraft, the lift required is greater as a result of the intermediate M-H aircraft burning more chemical fuel. With this plus the requirement of 300 feet per minute climb capability, the thrust required at cruise is higher for the low M-H aircraft. This powerplant weight increase is then due mainly to heavier engines and heat exchangers necessary because of the higher airflow rates required to generate the same thrust.

The net result is two airplanes not differing too much in general appearance and having approximately the same payload.

The high M-H aircraft is more similar to present day jets than it is to the other two aircraft. It has a lower aspect ratio and a swept back wing. In addition, its design point condition requires much higher power levels than either of the other two and a heavier structure. We can conclude that to gain altitude and Mach number capability is very costly in terms of payload (a decrease of 72 percent).

Engine Operating Conditions

The specific fan and compressor maps for the three aircraft are shown in figures 6(a) and (b) through 8(a) and (b). The design values used to operate on the generalized maps 5(a) and (b) are also indicated in table I. As can be seen from the design bypass ratios ranging from 8.8 to 3.7 and the fan diameter of 9.6 to 9.0 feet and the core diameter of 3.0 to 4.4 feet, these engines are somewhat larger in size than the engines of the Boeing 747 and Lockheed C5A.

On these maps are indicated the operating lines on full nuclear power without chemical augmentation. Since the nozzles are not choked and thrust does not equal drag at the design point, the design point does not fall on this operating line. Operating at thrust equals drag does not appear to offer any significant operating problems such as surge or overspeed except when the altitude rises significantly above the design value. The low design turbine-inlet temperatures (1700° R) do not rise significantly, and overheating or turbine cooling does not present a problem either. In fact, these operating lines appear to be quite typical of turbofan operation. Although the operating lines are not extended to takeoff conditions, this is due to the requirement of takeoff on chemical fuel only and not to inability to operate at this point.

Actual fan airflow rate varies from 1340 to 1050 lb/sec and compressor airflow rate from 137 to 222 lb/sec from the low M-H to the high M-H aircraft.

Aircraft Flight Envelopes

Flight envelopes for the three design aircraft are shown in figures 9 through 11. As can be seen from the figures, a region exists wherein thrust on full nuclear power is greater than the drag of the aircraft. Within this region then, the aircraft can fly on part power. The design point is located therein as a result of sizing the engines for 300 feet per minute climb capability at cruise. This region is bounded by a line where it is still possible to fly on nuclear power alone and where the full power thrust equals the drag. The engine operating conditions along this line were previously shown in figures 6(a) and (b) through 8(a) and (b).

By designing the engines for chemical augmentation, the flight envelope can be extended. Chemical fuel consumption rates have been indicated. It can be seen from the flight envelopes that to go to high Mach number at low altitude, a high chemical fuel rate is required. This high consumption rate relative to the payload of the aircraft implies that, if this capability is desired, the mission times will be very small when compared to the 10 000 hour lifetime of the reactor, heat exchangers, and engines.

This can be alleviated by increasing the payload by either designing the nuclear components for shorter lifetime, or increasing the gross weight of the aircraft. The effect of gross weight on payload was shown in reference 1 and of lifetime in reference 2. If either is changed, however, the aircraft should be reoptimized and the engine parameters will change. New flight envelopes would then have to be generated.

At the far right on the figures is a boundary beyond which aircraft drag exceeds engine thrust with full chemical augmentation. This boundary is a result of the high thrust requirements due to compressibility drag and allowing a maximum of a 20 percent overspeed of the engine compressor. Raising turbine-inlet temperature causes an increase in compressor speed. An arbitrary overspeed limit was assumed to be set by structural considerations; higher speeds would then require a new and heavier compressor and turbine design than that used herein. For the low M-H aircraft, surge of the compressor occurs before the overspeed limit is reached. Another boundary based on stall limitations also exists but has not been determined. The use of a variable-area primary exhaust nozzle might extend the flight envelope but has not been studied herein.

From figures 9, 10, and 11, we can also see that Mach number capability on nuclear power alone increases with altitude. The reasons for this will become apparent later.

Comparison of figures 9 and 10 indicates that a nuclear airplane designed for Mach 0.3 and 2000 feet is less desirable than the Mach 0.55, 20 000 foot airplane. The Mach 0.55 airplane not only covers the same range of Mach number and altitude on nuclear power only as the low M-H airplane but an even larger range. In addition, the payloads are almost the same.

Comparison of figures 10 and 11 shows that the Mach 0.8, 40 000 foot airplane has a much larger range of Mach number-altitude capability. However, this is achieved at a considerable cost in payload (72 percent).

Figure 12 shows air temperature out of the heat exchanger as a function of altitude for three different operating Mach numbers for this high M-H aircraft. The effect of altitude is much stronger than that of Mach number. As previously shown by the operating lines in figures 6(b) through 8(b), corrected airflow rate increases with increasing altitude. Since $\dot{W}_A \sqrt{\theta}/\delta$ is increasing at a slower rate than δ decreases $\dot{W}_A$ is decreasing. Therefore, since power is constant, ΔT of the air across the heat exchanger increases, giving higher turbine-inlet temperature on nuclear power alone. At the same time, the drag of the airplane decreases. Therefore, the burning of chemical fuel does not occur until the desired Mach number increases the drag above the thrust level provided by nuclear power alone. This is why higher Mach number is achievable on nuclear power only as the altitude increases.

CONCLUDING REMARKS

This analysis has indicated what capabilities are achieved and what losses are suffered as a consequence of picking a particular design point, based on the component performance employed in references 1 and 2.

Operation on nuclear power only was not restricted to a small area near the design point. In addition a larger envelope could be obtained by the use of chemical augmentation at a cost in endurance.

The flight envelopes demonstrated that an intermediate Mach number-altitude aircraft is attractive in that it can carry the same payload as a low M-H airplane an do it over a much larger range of flight conditions. Designing for a still larger flight envelope, hence greater versatility, incurs a heavy (72 percent) penalty in payload capability.

Engine operational requirements do not appear to differ significantly from that of contemporary turbofans.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, May 27, 1969
789-50-01-01

APPENDIX A

SYMBOLS

L/D	lift to drag ratio at cruise
M-H	Mach number - altitude
N_1	speed of fan - rpm
N_2	speed of compressor - rpm
P_2/P_1	fan pressure rise ratio
P_3/P_2	compressor pressure rise ratio
$\Delta P/P$	$P_2/P_1 - 1$ or $P_3/P_2 - 1$
ΔT	air temperature rise across heat exchanger - $^\circ$ R
WAFC	corrected fan airflow = $\dot{W}_{A_F} \sqrt{\theta_1}/\delta_1$ - lb/sec
WACC	corrected compressor airflow = $\dot{W}_{A_C} \sqrt{\theta_2}/\delta_2$ - lb/sec
$\dot{W}_{A_C}$	actual compressor airflow - lb/sec
$\dot{W}_{A_F}$	actual fan airflow - lb/sec
δ	ratio of total pressure at entrance to component to atmospheric pressure at standard sea level
η	efficiency, 0.88 design for fan, 0.86 design for compressor
θ	ratio of total temperature at entrance to component to atmospheric temperature at standard sea level

Subscripts

1	conditions at entrance to fan
2	conditions at exit of fan - compressor entrance
3	conditions at exit of compressor

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6. McKinney, John S.: Simulation of Turbofan Engine. Part 1: Description of Method and Balancing Technique. AFAPL-TR-67-125, Wright-Patterson AFB, Nov. 1967. (Available from DDC as AD-825197.)
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TABLE I. - DESCRIPTION OF DESIGN POINT AIRCRAFT CHARACTERISTICS

Altitude - feet	2000	20 000	40 000
Mach number	0.3	0.55	0.8
Payload - lb	155 500	153 000	43 000
Length - feet	264	264	264
Span - feet	301	257	264
Aspect Ratio	7.53	7.96	5.26
Wing Loading - lb/ft^2	82.7	118.9	72.9
Leading Edge Sweep	4.1°	3.9°	23.1°
Thickness to Chord	0.18	0.18	0.12
Power - MW	200	212	257
Thrust Required - lb	70 300	65 300	67 500
Shield Weight - lb	298 000	306 000	326 000
Fan Pressure Ratio	1.144	1.221	1.450
Compressor Pressure Ratio	6.110	10.333	8.001
Heat Exchanger Pressure Ratio	0.926	0.919	0.899
Turbine Inlet Temperature °R	1687	1711	1657
Bypass Ratio	8.763	6.563	3.708
Air Flow Rate/Engine - lb/sec	1340	1470	1050
Helium Flow Rate/Engine - lb/sec	30.0	32.0	37.7
Fan Diameter - feet	9.37	8.99	9.61
Core Diameter - feet	3.00	3.27	4.43
Core Airflow Rate/Engine-lb/sec	137	194	222
Cruise L/D	18.0	17.4	15.9
Total Powerplant Weight - lb	458 100	481 000	561 800
Total Structure Weight - lb	383 400	363 000	392 200

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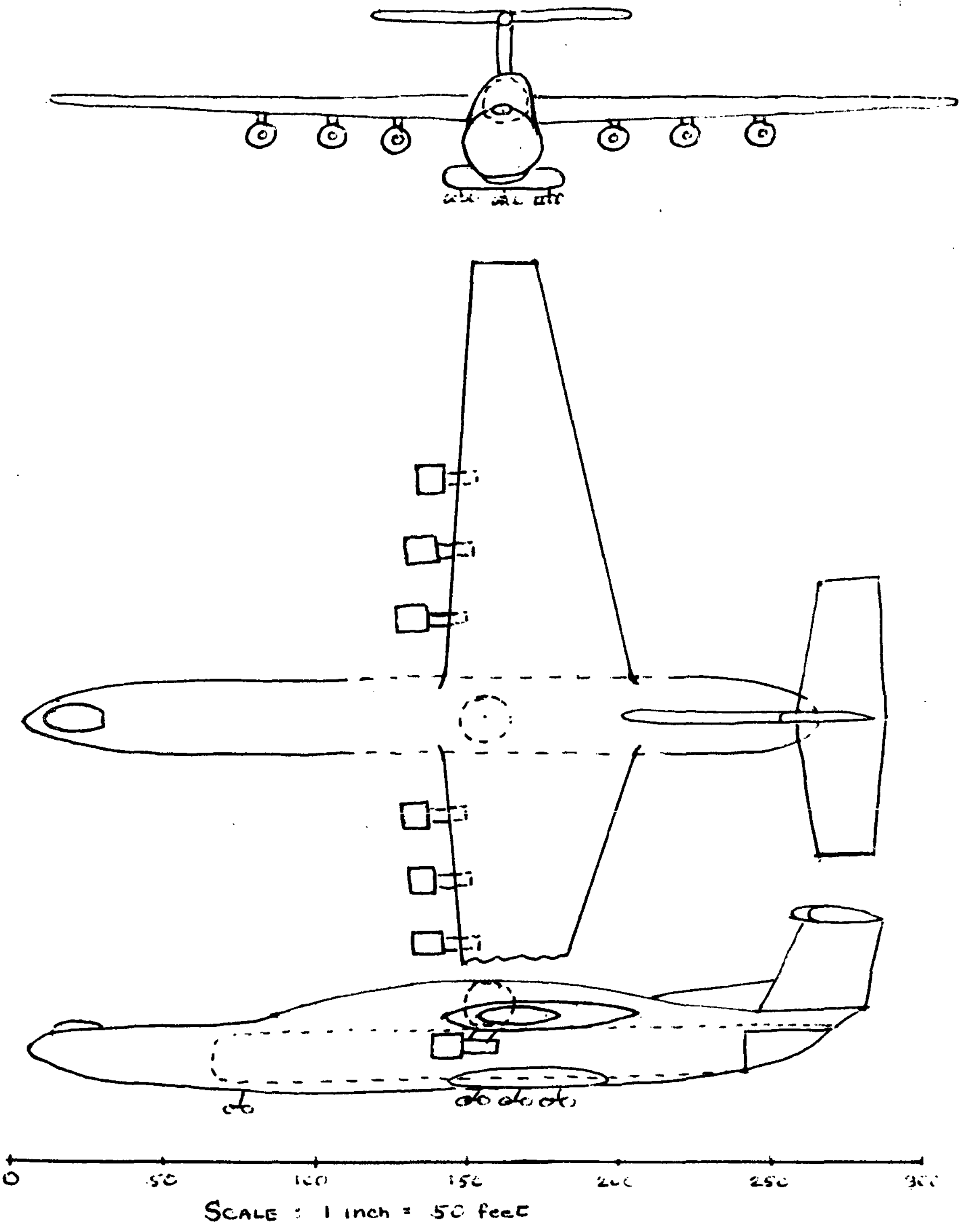


Figure 1. - Mach 0.3, 2000 feet design point airplane

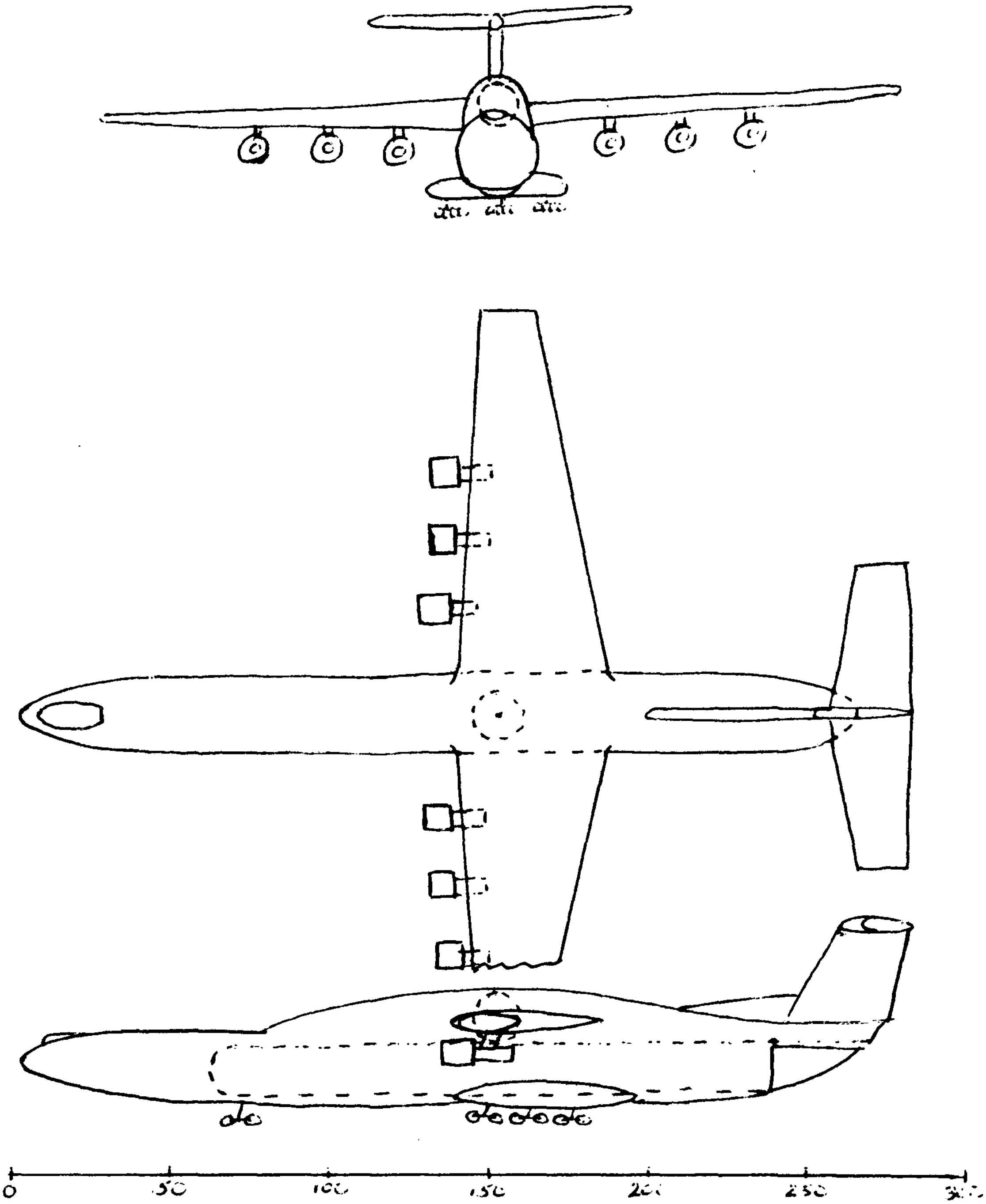


Figure 2. - Mach 0.55, 20 000 feet design point airplane

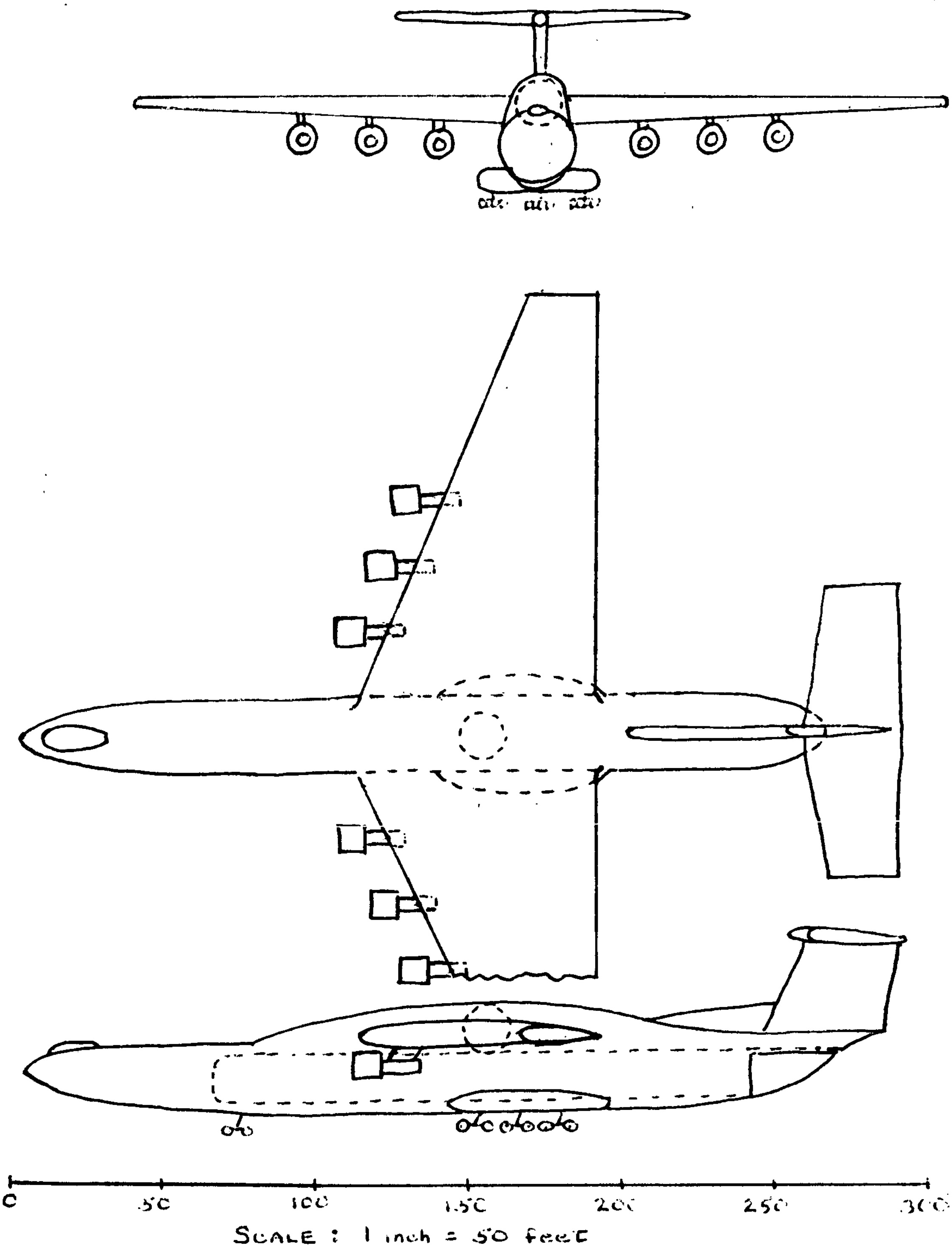


Figure 3. - Mach 0.8, 40 000 feet design point airplane

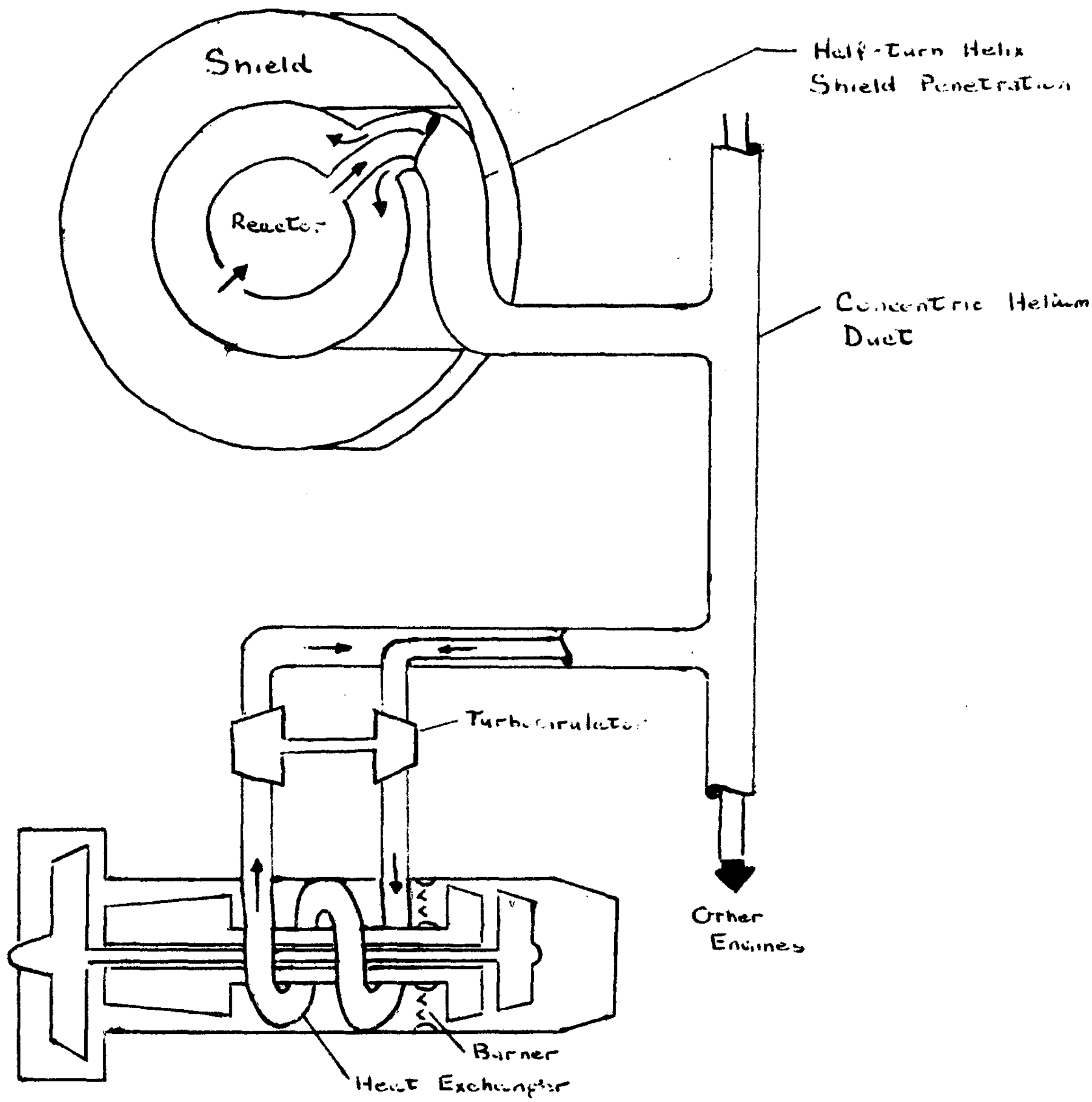
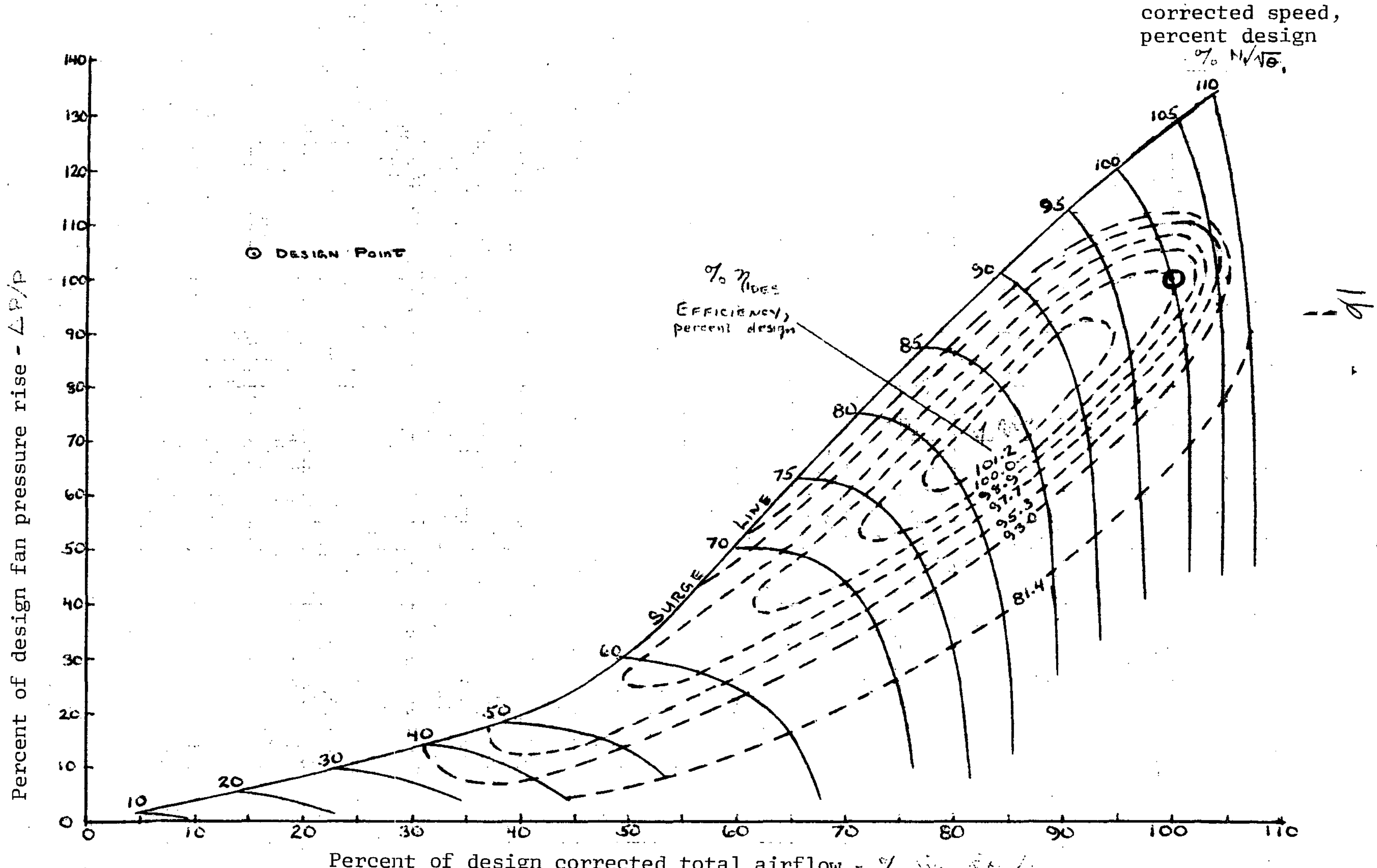
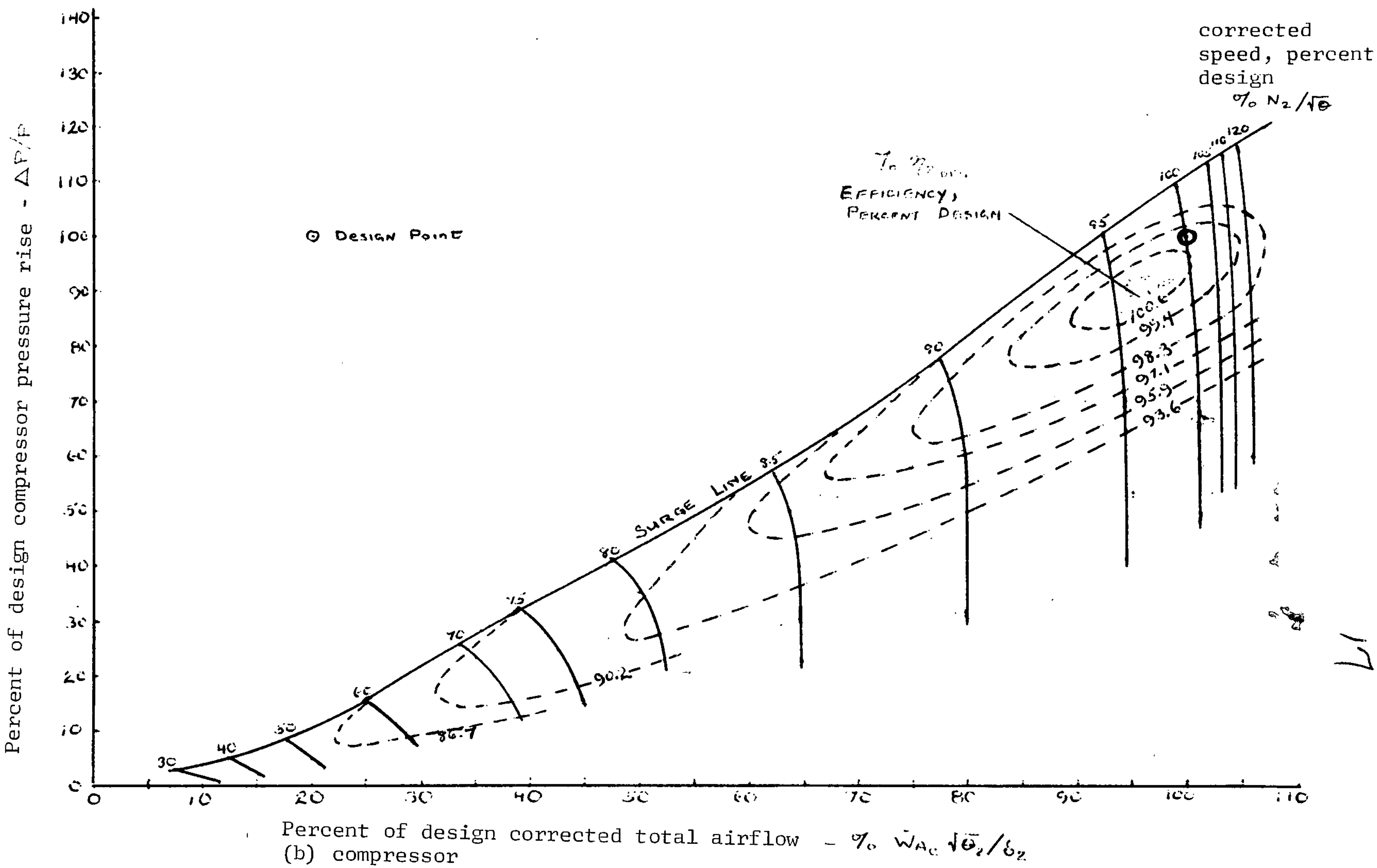


Figure 4. - Subsonic nuclear-powered airplane propulsion system



(a) fan

Figure 5. - Generalized Maps



(b) compressor

Figure 5. - Concluded

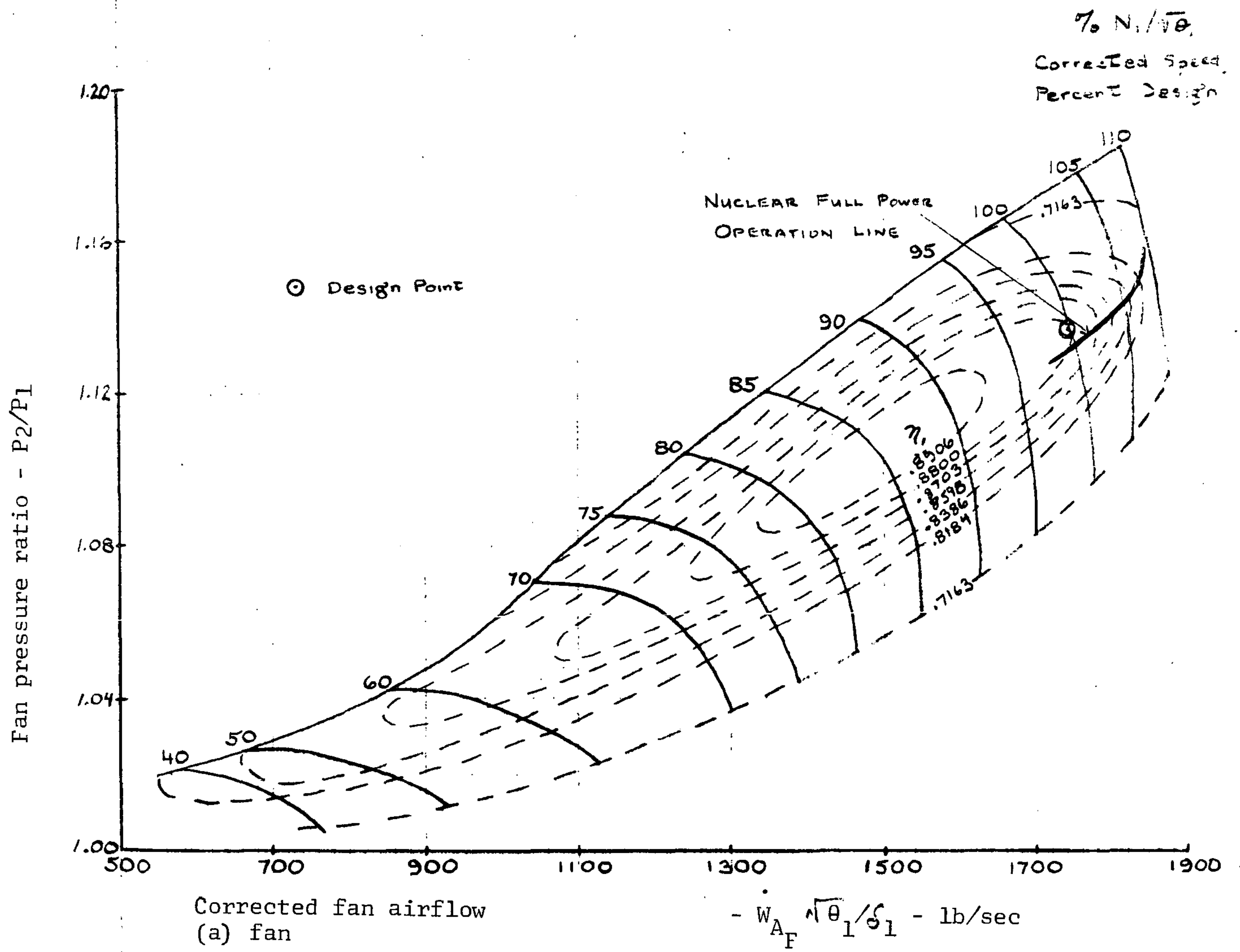
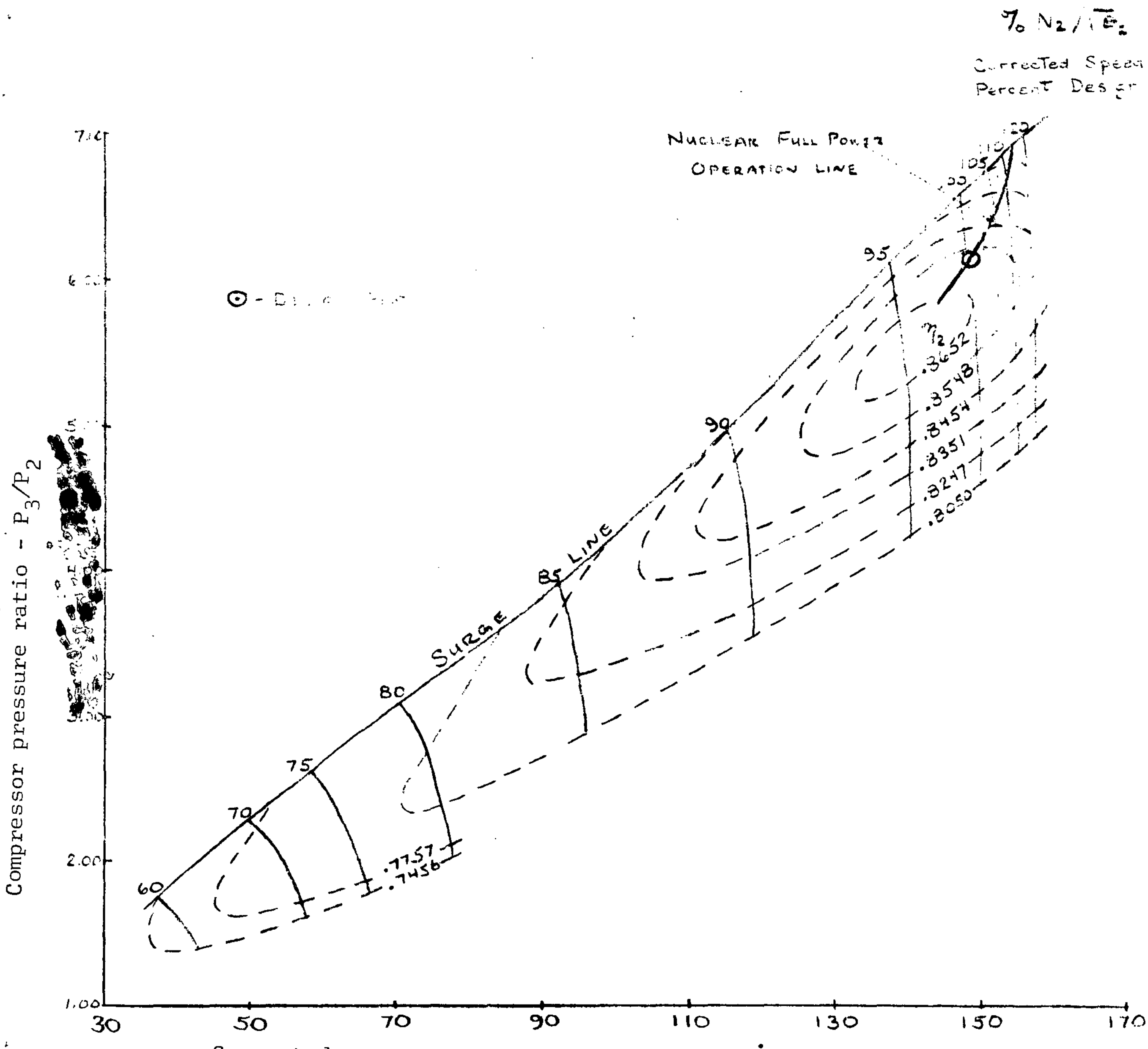


Figure 6. - Maps and operating lines on nuclear power only. Design point Mach 0.3, 2000 feet.



Corrected compressor airflow - $\dot{W}_{A_C}\sqrt{\theta_2}/\delta_2$ - lb/sec
(b) compressor

Figure 6. - Concluded

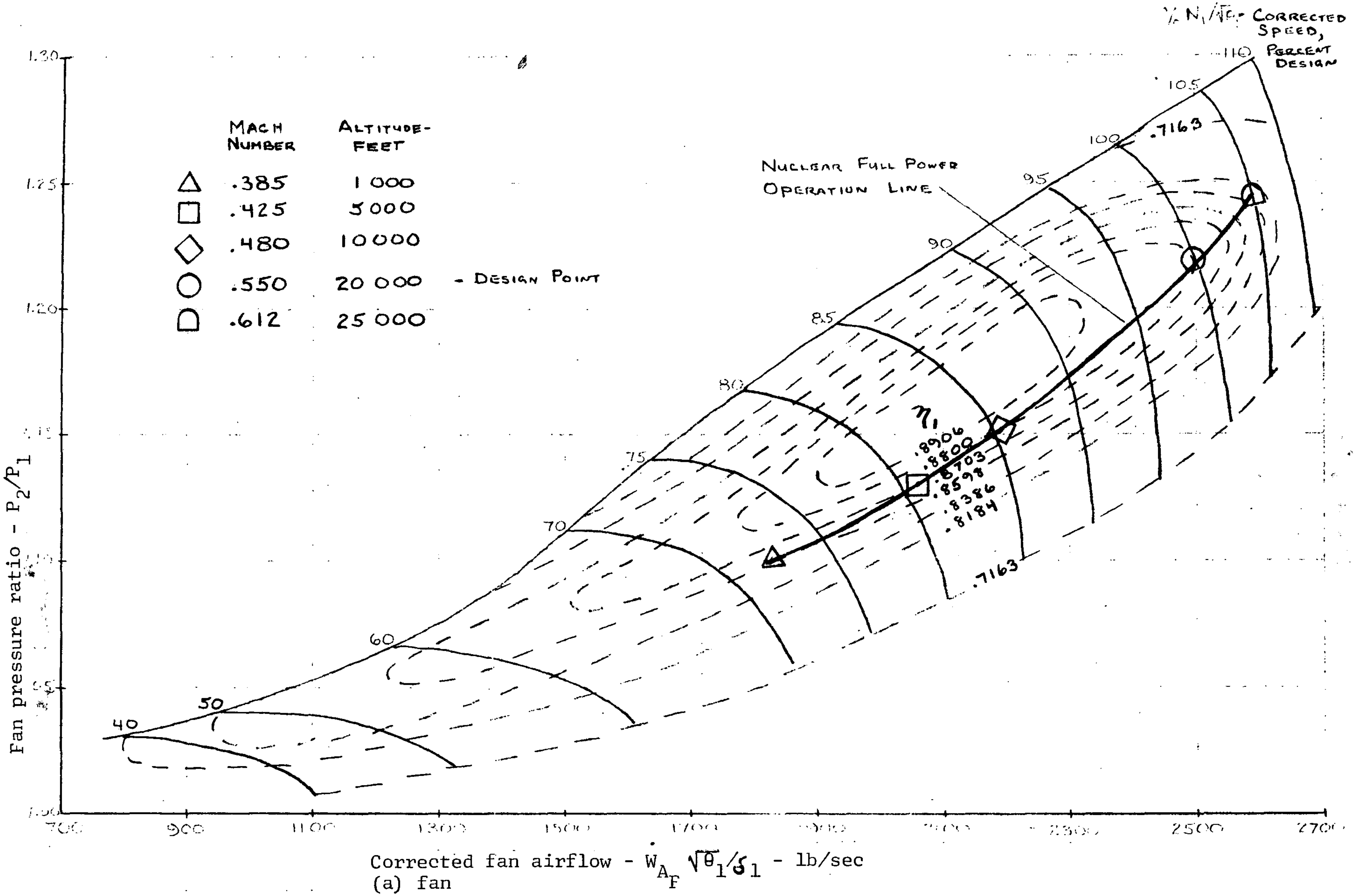


Figure 7. - Maps and operating lines on nuclear power only. Design point Mach 0.55, 20 000 feet.

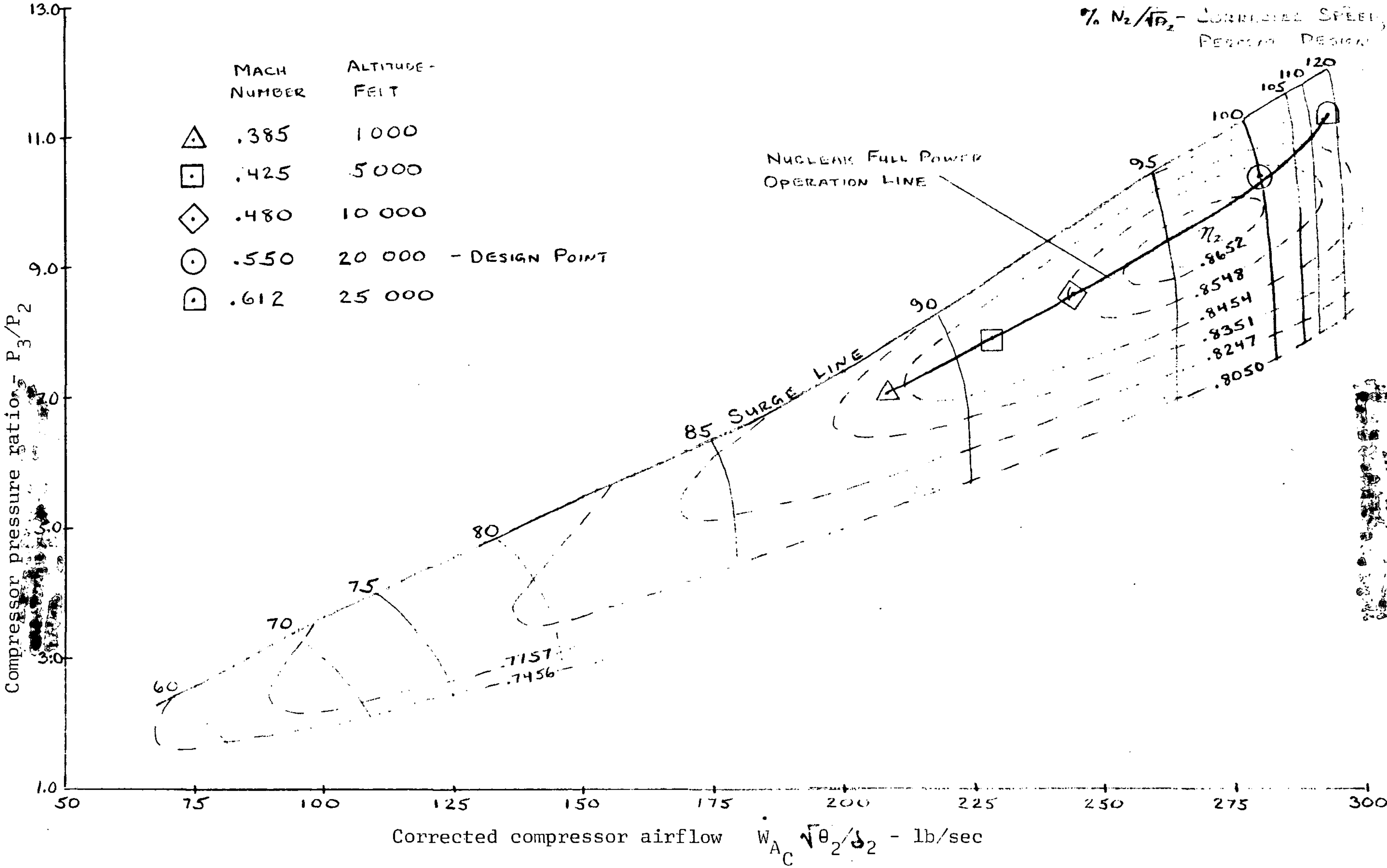
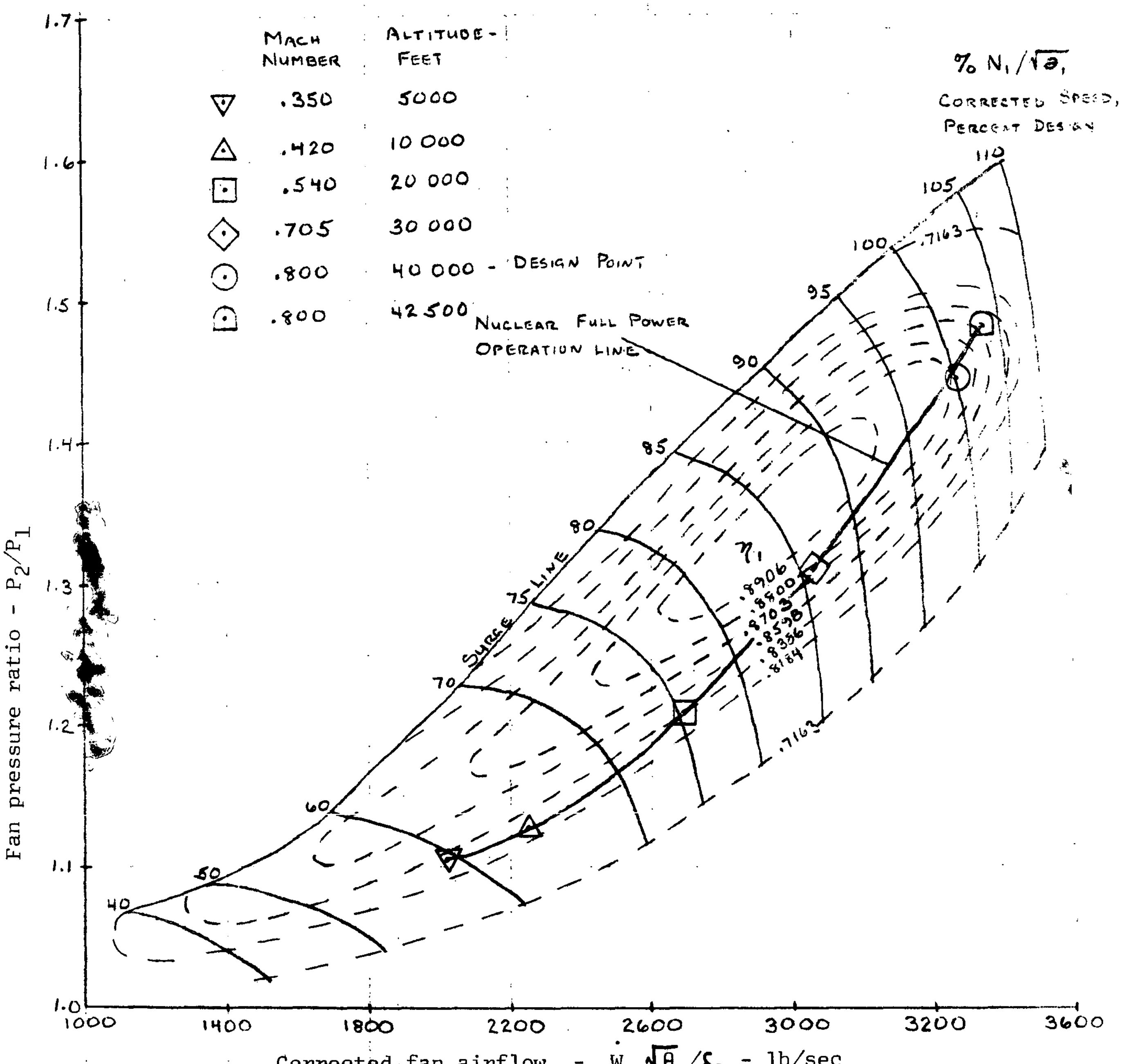


Figure 7. - Concluded



Corrected fan airflow - $\dot{W}_{A_F}\sqrt{\theta_1}/\delta_1$ - lb/sec

(a) fan

Figure 8. - Maps and operating lines on nuclear power only Design point Mach 0.8, 40 000 feet

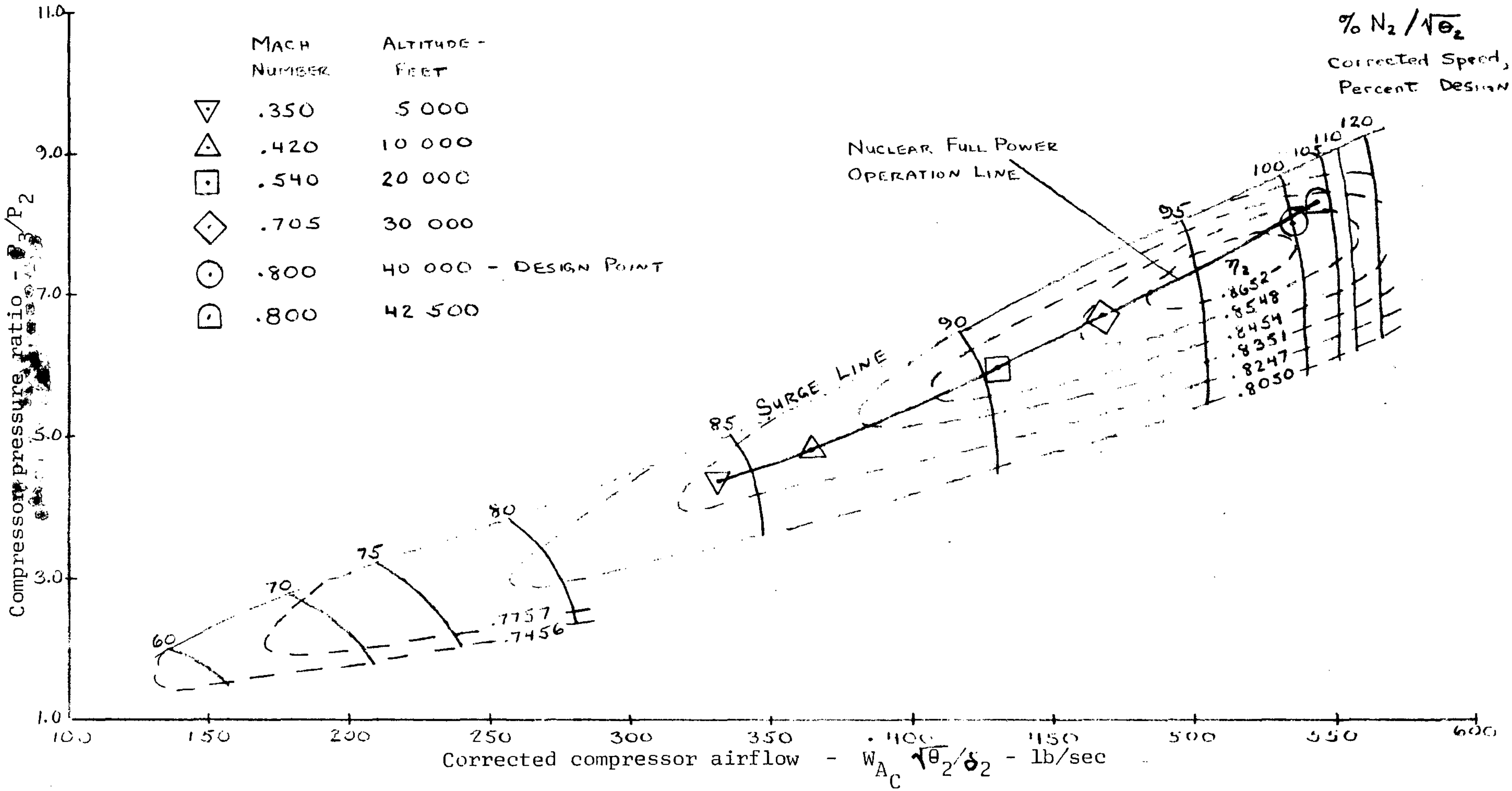


Figure 8. - Concluded

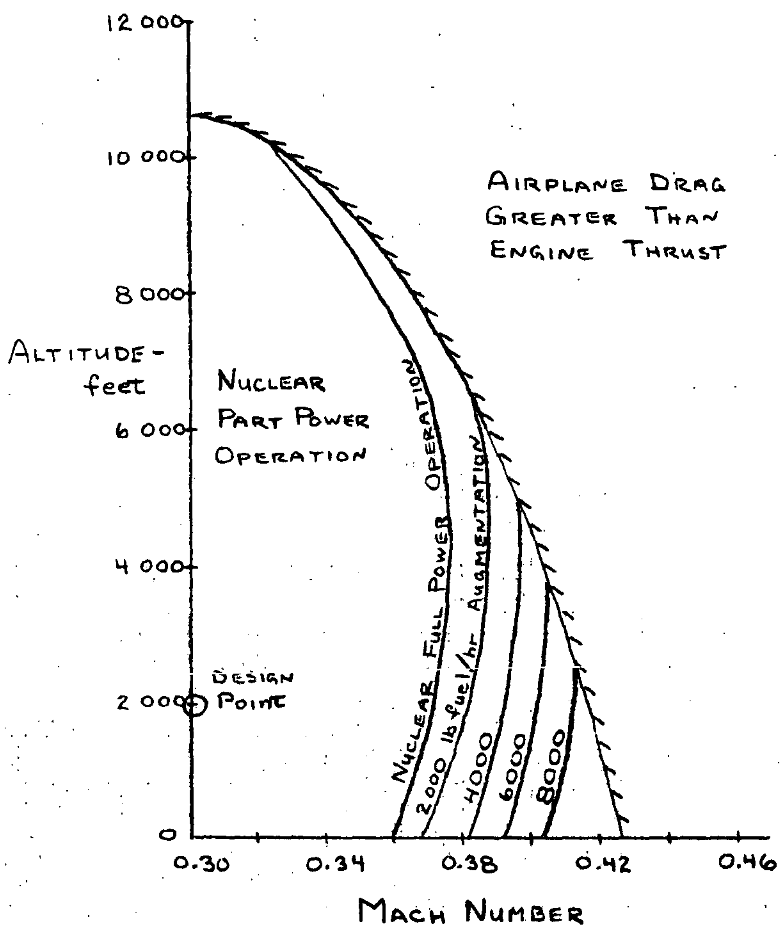


Figure 9. - Flight envelope; design point - Mach 0.3, 2000 feet, 1 000 000 lb gross weight, 10 000 hours life, payload 156 000 lb

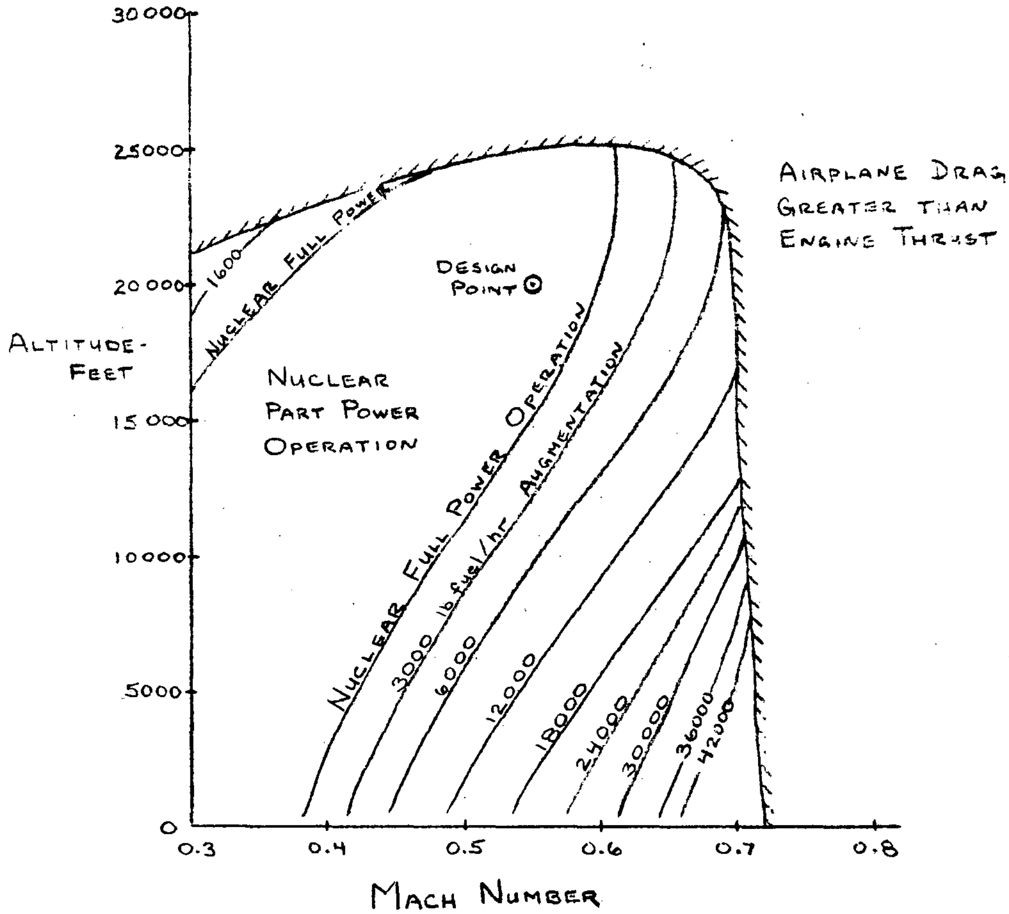


Figure 10. - Flight envelope; design point - Mach 0.55, 20 000 feet, 1 000 000 lb gross weight, 10 000 hours life, payload 153 000 lb

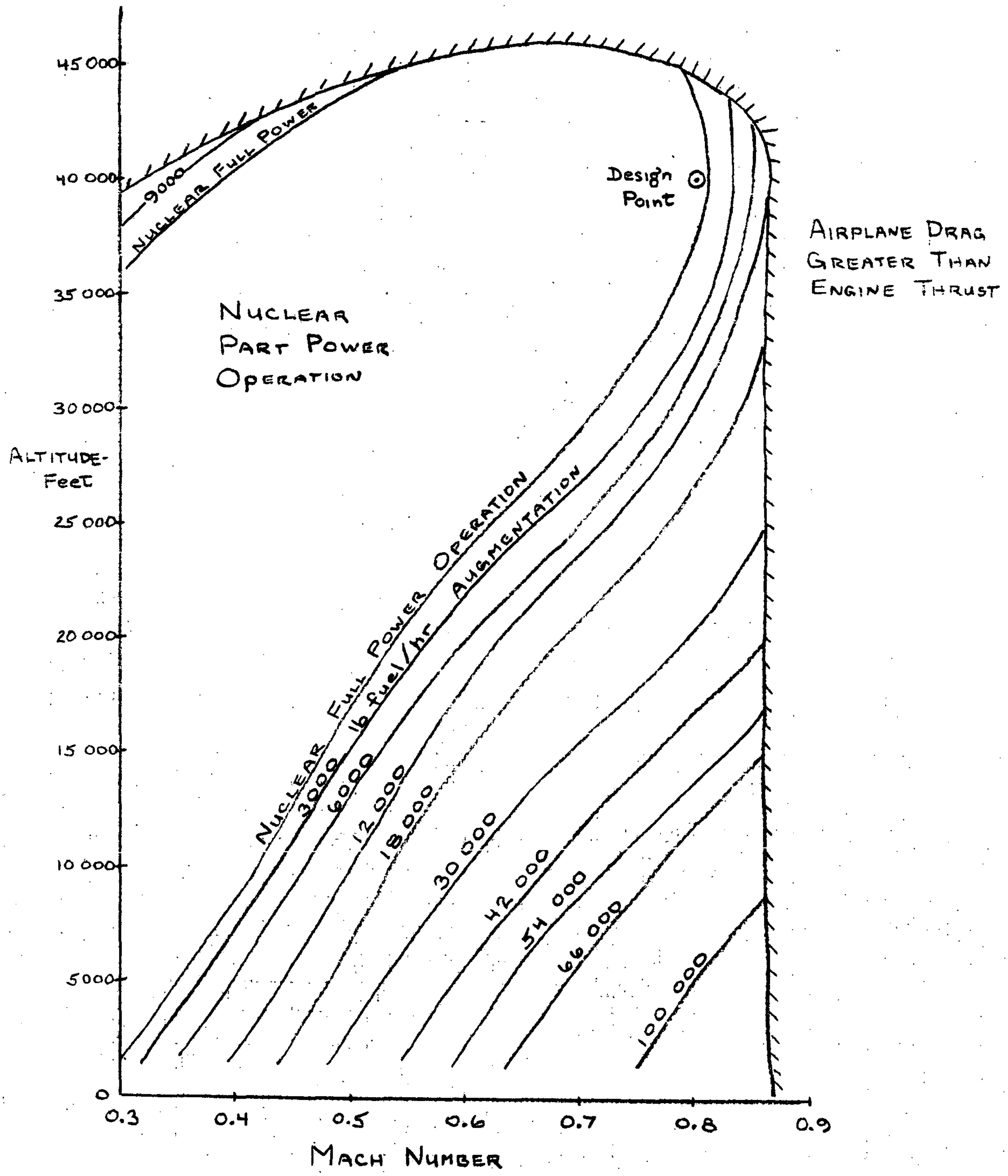


Figure 11. - Flight envelope; design point - Mach 0.8, 40 000 feet, 1 000 000 lb gross weight, 10 000 hours life, payload 43 000 lb

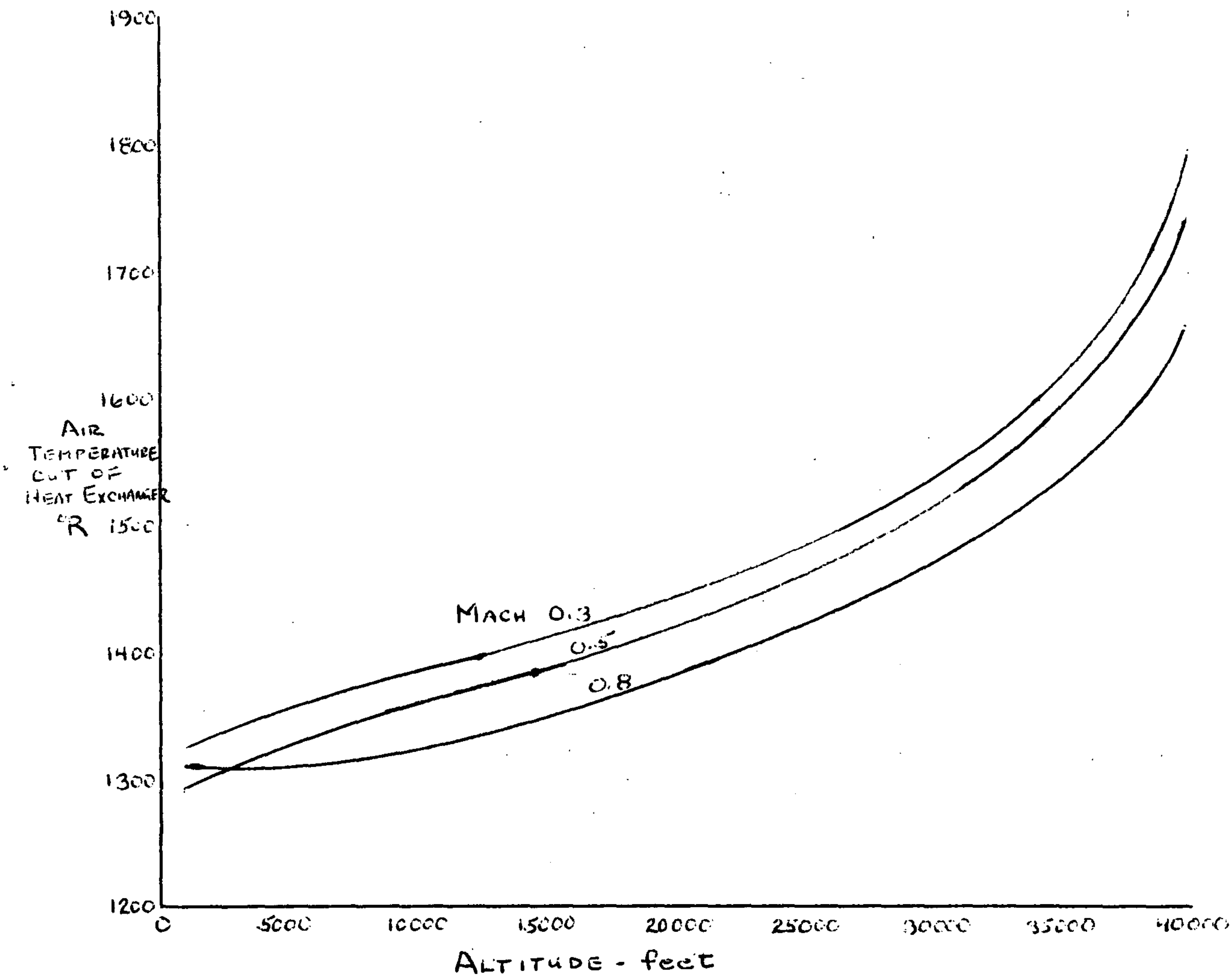


Figure 12. - Effect of altitude on air temperature out of heat exchanger; Mach 0.8, 40 000 feet altitude design point